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TPS CHALLENGES FOR NEPTUNE AEROCAPTURE

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A study to develop a conceptual design for an aerocapture mission at Neptune was conducted by a NASA systems analysis team comprised of technical experts from several NASA centers. Multidisciplinary analyses demonstrated that aerocapture could be accomplished at Neptune with a rigid aeroshell with a flattened ellipsled geometry flying at a nominal angle-of-attack of 40 degrees entering the Neptune atmosphere at an inertial entry velocity of ≈ 29 km/s. Aerothermal analyses demonstrated that both the peak convective and radiative heating rates in the stagnation region are very severe. Furthermore, due to the duration of the aerocapture trajectory, the total integrated heat loads are extremely large. TPS sizing analyses were conducted for a limited range of candidate TPS materials since such high peak heat fluxes limit candidate materials to dense, carbonaceous ablators. On the windward side, in regions away from the stagnation region, lower density ablators may suffice. Low-density ablators also are viable candidates on the lee side. However, there are significant uncertainties associated with the turbulent and radiative heating rates. TPS mass requirements for this mission are very large, and the ablator thickness requirements may be beyond current manufacturing capabilities.

NOMENCLATURE

σ	Standard deviation
ΔV	Change in velocity
A	Vehicle cross-sectional area
ARA	Applied Research Associates
ARC	NASA Ames Research Center
c.g.	Center-of-gravity
C_A	Axial force coefficient
C-C	Carbon-carbon
C_D	Vehicle drag coefficient
CFD	Computational Fluid Dynamics
C_N	Normal force coefficient
CP	Carbon phenolic
DOD	Department of Defense
EJGA	Earth-Jupiter gravity assist
EOL	End-of-life
JSC	NASA Johnson Space Center
L/D	Lift-to-drag ratio
LaRC	NASA Langley Research Center
LMA	Lockheed-Martin Astronautics
M	Vehicle mass
M/C_DA	Ballistic coefficient, kg/m^2
SCBA	Strip Collar Bonding Approach
SEP	Solar Electric Propulsion
TPS	Thermal Protection System
VJGA	Venus-Jupiter gravity assist

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INTRODUCTION

Through detailed trade studies,[1] the mission analysis team determined that aerocapture at Neptune could be accomplished with a rigid aeroshell configured as a flattened ellipsled with a lift-to-drag (L/D) ratio of ≈ 0.80 and a ballistic coefficient (M/C_DA) of ≈ 895 kg/m^2. The configuration is illustrated in Figure 1.

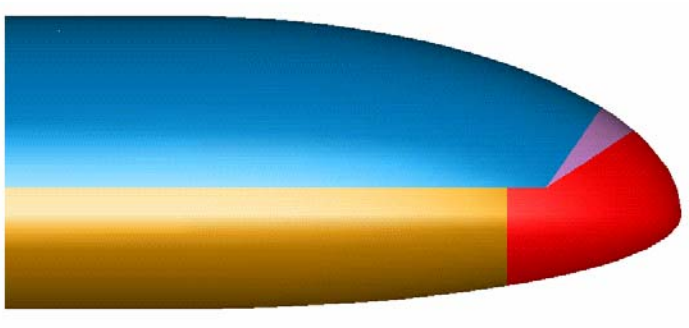

Figure 1. Flattened ellipsled geometry (2.88 m).

A mission analysis study[2] evaluated options to deliver an orbiter and two probes to Neptune. It was concluded that chemical trajectories exist that can deliver the required mass for an orbiter and two probes (~ 2000 kg) using aerocapture at Neptune but would require an Earth-Jupiter gravity-assist (EJGA) and the Delta IV H booster. EJGAs would necessitate an auxiliary chemical propulsion "stage" providing 0.5-3 km/s post launch deep space maneuvers. This study also concluded that trajectories also exist that can deliver the required mass for an orbiter and two probes using solar electric propulsion (SEP) and aerocapture. These options would use a Venus-Jupiter gravity-assist (VJGA) and the Delta IV H. The inertial entry velocity for these trajectories is around 28-30 km/s for 10-11 years flight time. It was concluded that a reasonable mission design (in terms of maximizing delivered mass while minimizing trip time) could be accomplished with a launch on February 17, 2017 on a Delta IV H with a total launch mass of 3298 kg dry (4735 kg wet). The mission would use a 6-engine SEP system (30 kW, EOL) and a VJGA. This would be a 12.25 year Neptune Orbiter Mission with a 10.25-year trip time and 2 (or more) years in Neptune Orbit. The payload would include two identical probes and would use aerocapture to establish a target orbit of 430,000 km x 3,986 km (which would enable Triton observations), with a 157.3 degree inclination, retrograde.

The atmosphere around Neptune is composed primarily of hydrogen and helium with small concentrations of methane and nitrogen. The nominal atmospheric composition is approximately 80% H_2, 19% He, and 1% CH_4. Justus et al[3] developed a model for the Neptune atmosphere (Neptune-GRAM) based on data from Voyager and elsewhere. Allowing for measurement uncertainties and the expected latitudinal, altitudinal, seasonal and diurnal variations, the model accounts for a significant variation in atmospheric density with altitude, as shown in Figure 2.

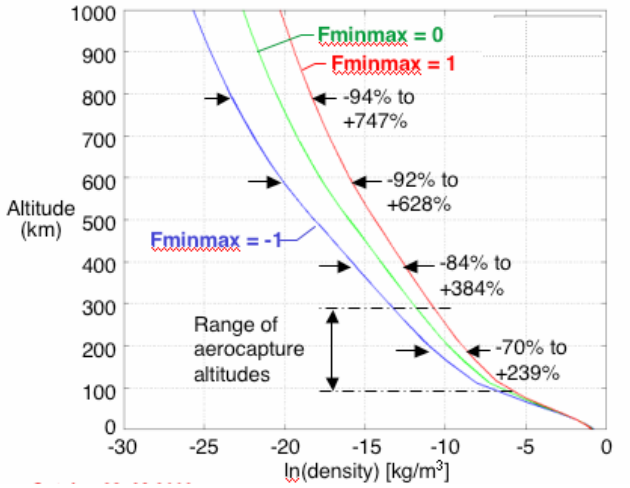

Figure 2. Neptune-GRAM engineering models for the density of the Neptune atmosphere.

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Relatively large measurement uncertainties are represented within a prescribed envelope of minimum-average-maximum density versus altitude. A single model input parameter (Fminmax) allows users of Neptune-GRAM to select where within the min-max envelope a particular simulation will fall. Fminmax = -1, 0, or 1 selects minimum, average, or maximum conditions, respectively, with intermediate values determined by interpolation. The percentages shown at selected altitudes in Figure 2 indicate the range of density uncertainty between the minimum density model and the maximum density model in comparison to the average.

An evaluation of the aerodynamic stability[4] of the flattened ellipsled was conducted. The aerodynamic database was developed from viscous LAURA CFD[5,6,7] solutions. A preliminary stability analysis showed that the flat-bottom ellipsled is longitudinally and laterally stable. An assessment of the aerodynamic uncertainties was based on a JSC ellipsled analysis for Mars, consistent with the X-33 aerodynamic database uncertainty model (Ref. NASA TP-1998-206544).

Using these results for vehicle aerodynamics, a Neptune aerocapture simulation and Monte Carlo analysis approach[8] was developed. Three sets of uncertainties were included in the Monte Carlo analysis: delivery errors in initial states, atmospheric variability (mean range and perturbations) and several aerodynamic factors – C_A, C_N, trim angle of attack (pitching moment), c.g. uncertainty. The parameters are each randomly varied over a specified range and distribution. More than 2000 trajectory simulations are completed in one Monte Carlo analysis and the results are used for aerocapture performance statistics to determine robustness, margin and risk, guidance development, stress case identification (control algorithm development to be looked at in the future), and statistical distributions of critical parameters and design trajectories for sub-system design. The analysis demonstrated that 100% of possible aerocapture trajectories successfully achieve aerocapture, as seen in Figure 3. Furthermore, a 360 m/sec 3-σ ΔV would be required to raise the periapsis to 3986 km and correct the apoapsis to 430,000 km. The results were used to select a design trajectory for the evaluation of aerodynamic heating and associated TPS performance. For this purpose the results for heat load vs. heat rate were evaluated, as shown in Figure 4. Monte Carlo trajectory #1647 was selected for the reference concept TPS design since it provides the highest heat load trajectory of the all the considered cases and the heat rate is in the 98th percentile.

Figure 4 illustrates that the Monte Carlo trajectories are in the high heat rate, low heat load range of the lift up, lift down corridor. In this corridor, the peak heating rates[‡] range from a lift up maximum of 3155 W/cm^2 to a lift down maximum of 1122 W/cm^2. Similarly, the heat loads[‡] range from a lift up minimum of 185 kJ/cm^2 to a lift down maximum of 442 kJ/cm^2.

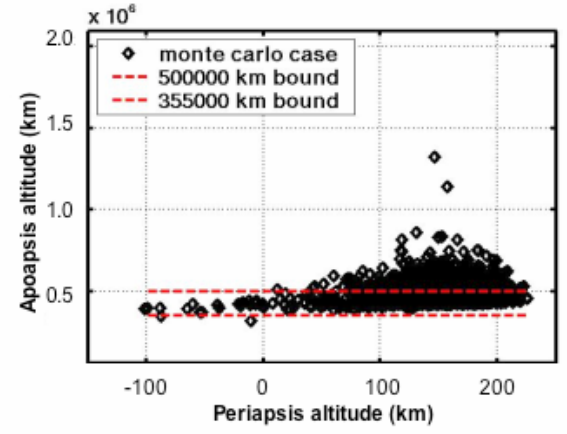

Figure 3. Monte Carlo analysis results for aerocapture trajectories

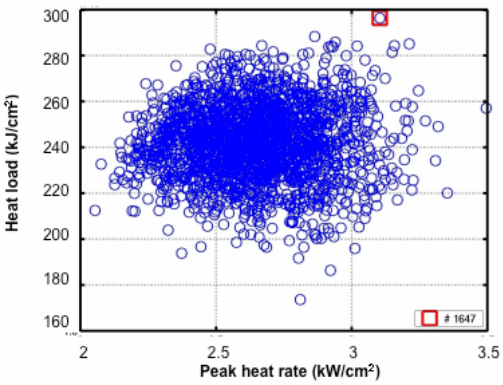

Figure 4. Heat load vs. heat rate from Monte Carlo results

‡ The heat rates and heat loads shown are simply used as an aeroheating indicator and actually are for convective stagnation point only and calculated for a 1m-nose radius.

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CANDIDATE TPS MATERIALS

Early in the study, NASA Ames did preliminary analyses to estimate the anticipated heating environment for Neptune aerocapture. For that purpose, Ames evaluated the flattened ellipsled geometry on the limiting lift up (undershoot) trajectory with a minimum density atmosphere model. At the time, the preliminary vehicle design had a ballistic coefficient of 400 kg/m^2 in contrast to 895 kg/m^2 for the final design.

Convective heating was calculated through three-dimensional CFD solutions with the DPLR[9] code and radiative heating was calculated with the NEQAIR[10] code using an 11 species model. Figure 5 illustrates heating rates at selected locations on the body at a time in the trajectory when peak heating is experienced. The first numbers are radiative heating rates and the adjoining numbers are convective heating rates. On the basis of these values candidate TPS materials were defined for various regions on the body.

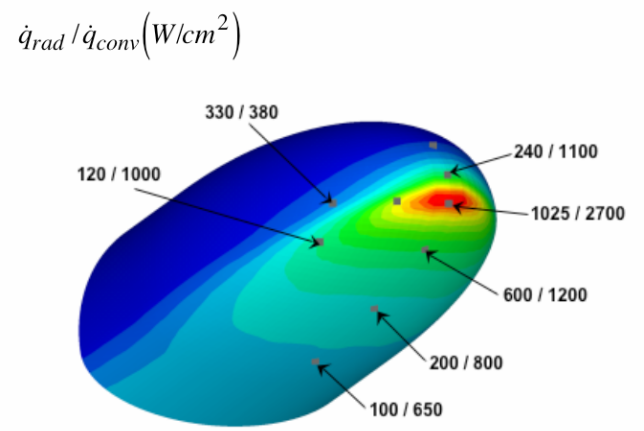

Figure 5. Preliminary radiative and convective heating rates on lift up trajectory.

In the windside stagnation region of the vehicle, where heating rates are the highest, candidate materials and their characteristics are summarized in Table 1. Fully dense carbon phenolic is a robust, mature, and well-characterized material that is capable of handling high heat fluxes and pressures. It was used by NASA as the forebody TPS on the Pioneer Venus and Galileo probes and as rocket nozzles for solid propellant boosters. It is well suited for severe heating environments where significant ablation would be anticipated. Its density and composition prevent it being a very good thermal insulator and, as a result, it would be a heavy TPS solution for more modest environments. Reduced density carbon phenolic is a *notional* material based on retaining the good ablation performance associated with a carbon reinforced phenolic composite but improving its thermal insulation performance by reducing the composite density. Such materials were briefly studied in the 1980s and a few composites were fabricated and tested with encouraging results. The Genesis TPS is comprised of a thin carbon-carbon facesheet attached (through co-processing) to low-density carbon fiberform insulation. It was manufactured as a single-piece heatshield for the 60° cone-shaped forebody of the 1.5 m diameter Genesis probe. The material was qualified through arc jet testing to heat fluxes as high as 700 W/cm^2 for the Genesis mission. Because of its all-carbon construction, it is not the best thermal insulator. The modified Genesis TPS is also a *notional* material, in which part of the carbon fiberform insulation is replaced with a low-density ceramic tile to improve its thermal insulation performance.

For the windside expansion region, where heating rates are somewhat lower, candidate materials are summarized in Table 2. Many of the same materials that are candidates for the windside stagnation region are also viable candidates for the windside expansion region. The exception is fully dense carbon phenolic, which has been replaced with PICA (Phenolic Impregnated Carbon Ablator), in which a low-density carbon fiberform is impregnated with phenolic resin. PICA is being used as the forebody TPS on the 60° cone-shaped forebody of the 0.83 m diameter Stardust probe. It has been tested to heat fluxes in excess of 1500 W/cm^2 with good ablation performance, but begins to exhibit char spall at pressures in excess of 0.6 atm.

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Table 1. Candidate TPS materials for windside stagnation region

Material	Fully-dense CP	Reduced density CP	Genesis	Modified Genesis
Density (g/cm^3)	1.45	0.967	1.8/0.18	1.8/0.18/0.192
Description	Fully-dense tape-wrapped or chopped molded heritage material (used on Pioneer-Venus and Galileo probes).	Reduced density tape-wrapped or chopped-molded composite (studied in the 70s and 80s).	Carbon-carbon facesheet over carbon fiberform insulator.	Carbon-carbon facesheet over carbon fiberform insulator over low-density ceramic insulator (e.g., AETB 20/12).
Optical Properties				
Solar absorptance	≈ 0.90	≈ 0.90	≈ 0.90	≈ 0.90
Total hemis. emittance	≈ 0.90	≈ 0.90	≈ 0.90	≈ 0.90
Performance limits	Ablative. No oxidation in Neptune's H_2-He environment, but will sublime at high heat fluxes. Will spall at very high heat fluxes (> 25 kW/cm^2).	Ablative. No oxidation in Neptune's H_2-He environment, but will sublime at high heat fluxes. Will spall at very high heat fluxes (< 25 kW/cm^2) and pressures).	Ablative. No oxidation in Neptune's H_2-He environment, but will sublime at high heat fluxes. Robust ablator, but could spall at very high heat fluxes, depending upon C-C processing.	Ablative. No oxidation in Neptune's H_2-He environment, but will sublime at high heat fluxes Robust ablator, but could spall at very high heat fluxes, depending upon C-C processing.
Uncertainties	A very poor insulator. Best application is for environment where significant ablation is expected. Difficult to fabricate tape-wrapped construction in thicknesses > 2 inches.	Better insulator than fully dense. Spall threshold = $f(\dot{q}, p)$ unknown and (most likely) a function of composite density	Genesis concept only qualified to ≈ 700W/cm^2 but should be capable of much higher fluxes. Fabricate as 1-piece or tiles? Issue is (likely) expansion compatibility	Same issues as Genesis with added complexity of bonding a 3rd material.

Finally, for the leeside region, where heating rates are much lower and radiative heating is estimated to be negligible, candidate materials are summarized in Table 3. SLA-561V is a silicone-based ablative in honeycomb developed by Lockheed-Martin (LMA) and is a flight-proven material that was used as the forebody heatshield on the Mars Viking, Mars Pathfinder and Mars Exploration Rover entry probes. The SRAM17 and SRAM20 materials are also silicone-based and were developed by Applied Research Associates (ARA). They are manufactured by a different technique, called the Strip Collar Bonding Approach (SCBA). PhenCarb20 is another ARA material manufactured by the SCBA technique, which uses a phenolic resin rather than silicone. It is capable of handling higher heat fluxes than the silicone-based materials, at the sacrifice of insulation efficiency.

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AEROTHERMAL ENVIRONMENTS

A team from NASA's Langley Research Center (LaRC) and NASA's Ames Research Center (ARC) who

Table 2. Candidate TPS materials for windside expansion region

Material	PICA	Reduced density CP	Genesis	Modified Genesis
Density (g/cm^3)	0.240	0.967	1.8/0.18	1.8/0.18/0.192
Description	Low-density carbon fiberform partially filled with phenolic resin.	Reduced density tape-wrapped or chopped-molded composite (studied in the 70s and 80s).	Carbon-carbon facesheet over carbon fiberform insulator.	Carbon-carbon facesheet over carbon fiberform insulator over low-density ceramic insulator (e.g., AETB 20/12).
Optical Properties				
Solar absorptance	≈ 0.90	≈ 0.90	≈ 0.90	≈ 0.90
Total hemis. emittance	≈ 0.80	≈ 0.90	≈ 0.90	≈ 0.90
Performance limits	Ablative. No oxidation in Neptune's H_2-He environment, but will sublime at high heat fluxes. Excellent low-density ablator, but not the best insulator. Will spall at high heat fluxes and pressures.	Ablative. No oxidation in Neptune's H_2-He environment, but will sublime at high heat fluxes. Will spall at very high heat fluxes (< 25 kW/cm^2) and pressures).	Ablative. No oxidation in Neptune's H_2-He environment, but will sublime at high heat fluxes. Robust ablator, but could spall at very high heat fluxes depending upon C-C processing.	Ablative. No oxidation in Neptune's H_2-He environment, but will sublime at high heat fluxes Robust ablator, but could spall at very high heat fluxes, depending upon C-C processing.
Uncertainties	Should be better insulator than reduced density CP. Spall threshold = $f(\dot{q}, p)$ unknown.	Better insulator than fully dense. Spall threshold = $f(\dot{q}, p)$ unknown and (most likely) a function of composite density.	Genesis concept only qualified to ≈ $700 W/cm^2$ but should be capable of much higher fluxes. Fabricate as 1-piece or tiles? Issue is (likely) expansion compatibility.	Same issues as Genesis with added complexity of bonding a 3rd material.

conducted independent flow field and radiative transport computations did the aerothermal environment definition of the flattened ellipsled geometry for Neptune aerocapture.[11] The aeroheating team held regular meetings to compare results and identify differences in thermochemical and radiation modeling, assumptions, boundary conditions, etc. Very good agreement was attained for laminar convective heating between LaRC's LAURA[5,6,7] code and ARC's DPLR[9] code. That was not true for turbulent heating where the LAURA code predicted significantly higher turbulent heating rates in comparison to DPLR.[11]

Radiative heating was calculated by LaRC using the RADEQUIL[12,13] code and by ARC using the NEQAIR[10] code. Again, there were significant differences, as discussed in Ref. 11. Based on these results, *estimates* of the heating

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environments along Monte Carlo trajectory #1647 were completed, and factors ("Low", "Med", "High") were applied to nominal predictions to represent the range of uncertainties. Transition to turbulence due to significant ablation resulting from high total heating rates is expected.

Table 3. Candidate TPS materials for leeside region

Material	SLA-561V	SRAM 17	SRAM 20	PhenCarb-20
Density (g/cm^3)	0.256	0.272	0.320	0.320
Description	Low-density cork silicone composite in Flexcore honeycomb.	Low-density cork silicone composite fabricated w/ strip collar bonding technique.	Low-density cork silicone composite fabricated w/ strip collar bonding technique.	Low-moderate density carbon fiber reinforced phenolic composite fabricated w/strip collar bonding technique.
Optical Properties				
Solar absorptance	≈ 0.50	≈ 0.50	≈ 0.50	≈ 0.50
Total hemis. emittance	≈ 0.78	≈ 0.78	≈ 0.78	≈ 0.78
Performance limits	Charring Ablator. No (little) recession at heat fluxes ≤ 100 W/cm^2. Char spall at pressures greater than ≈ 0.25 atm. Differential recession between matrix and honeycomb may cause BL transition.	Charring Ablator. No recession due to oxidation, but will sublime at high heat fluxes. Differential recession between composite and interface strips may cause BL transition.	Charring Ablator. No recession due to oxidation, but will sublime at high heat fluxes. Differential recession between composite and interface strips may cause BL transition.	Charring Ablator. No recession due to oxidation, but will sublime at high heat fluxes. Differential recession between composite and interface strips may cause BL transition.
Uncertainties	Data on recession rates at high heat fluxes (> 100 W/cm^2) relatively sparse.	Low density composite will probably spall. Spall threshold = $f(\dot{q}, p)$ unknown.	Low density composite will probably spall. Spall threshold = $f(\dot{q}, p)$ unknown.	Low density composite will probably spall. Spall threshold = $f(\dot{q}, p)$ unknown.

TPS SIZING

For purposes of evaluation, the vehicle was divided into 4 zones for TPS sizing as shown in Figure 6. The candidate TPS materials were selected and sized, with the FIAT[14] code, for the maximum heating point in each zone. The heatshield (forebody) is defined by zone 1 + zone 2. The backshell (aftbody) is defined by zone 3 + zone 4. Post-aerocapture aeroshell separation occurs between the heatshield and backshell.

For the heatshield (Zones 1 and 2), the material models employed with FIAT were extrapolated to heating conditions beyond the range for which they have been validated. The exception is fully dense carbon phenolic that has exhibited excellent performance at even more severe ground and flight conditions in development of the heatshield for the Galileo entry probe. In principle, the other material candidates should perform adequately at these conditions as similar materials have been tested with high-energy lasers at such conditions with good performance.

For the backshell (Zones 3 and 4), the material models employed with FIAT have been validated with ground

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test data at similar heating conditions and extrapolation was not required.

NOSE REGION (ZONE 1)

The "Low" and "Med" aeroheating rates and loads along Monte Carlo trajectory #1647 are shown in Figure 7. After further aeroheating analyses, it was concluded that the "High" values are outside the expected range. The "Med" level of aeroheating rates and loads was used for TPS sizing for the reference vehicle and is labeled "Reference" in all the following figures.

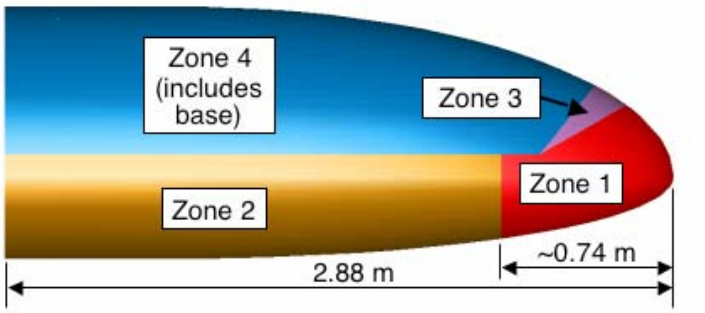

Figure 6. Zones selected for TPS evaluation.

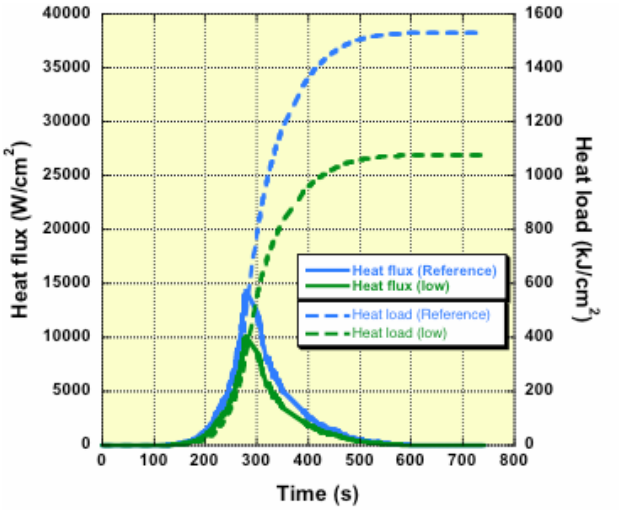

Figure 7. Total low and medium (Reference) heat rates and heat loads for nose region (Zone 1)

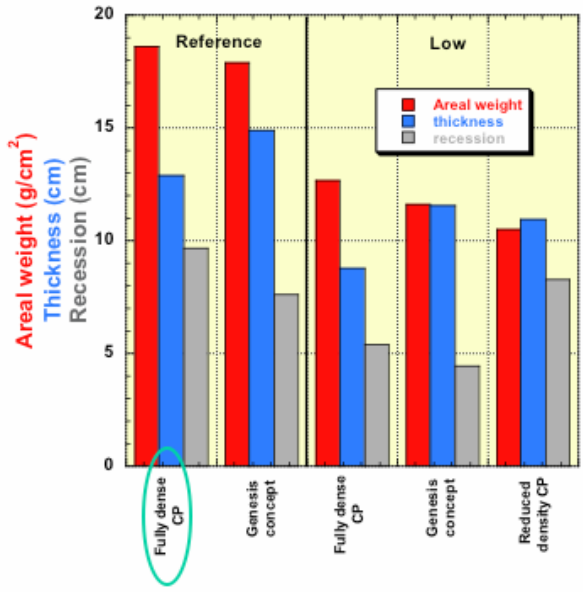

Figure 8. TPS sizing at the nose (Zone 1).

Note that the peak heat fluxes are in the range from 10-15 kW/cm^2. Only dense, carbonaceous materials can accommodate such high heat fluxes reliably. Furthermore, the total heat loads are in the range from 1076 to 1530 kJ/cm^2, significantly larger than the heat load for the most severe mission ever flown, i.e., Galileo. Consequently, for the reference ("Med") condition, only fully dense carbon phenolic (ρ = 1.45 g/cm^3) and the Genesis concept (carbon-carbon facesheet over carbon fiberform insulator) were considered. As shown in Figure 8, fully dense carbon phenolic (CP) is very heavy (≈ 18.6 g/cm^2) and very thick (≈ 12.9 cm). Furthermore, predicted surface recession is very large (≈ 9.6 cm). It is doubtful that tape-wrapped CP could be fabricated to these thicknesses with adequate quality. Such thicknesses were achieved for the nosecap on the Galileo probe, but that was a chopped molded construction of a relatively small nosecap. To fabricate a one-piece chopped molded heatshield for this large Neptune aerocapture vehicle would require presses of a size that do not currently exist. The option is a buildup of

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chopped molded tiles, but using tiles and the requisite gap fillers at such severe conditions introduces additional risk. In comparison, the Genesis concept has marginally lower areal weight (≈ 17.9 g/cm^2), but is even thicker (≈ 14.9 cm). The sizing analysis determined that the C-C facesheet needed to be ≈ 8.9 cm thick backed by ≈ 6 cm of carbon fiberform. Fabricating a cloth-reinforced carbon-carbon to that thickness with good uniformity and quality is a challenge. The predicted surface recession is also large (≈ 7.6 cm), but less than CP. The TPS requirements at "Low" condition heating are about 2/3rds those at the reference condition. It should be noted that for the evaluation of the "Low" condition, a reduced density carbon phenolic (ρ = 0.96 g/cm^3) was included in the evaluation. Such materials were fabricated and evaluated for other TPS applications in the 70s and 80s, but never used. The reduced density CP looks relatively competitive at the "Low" condition, but it is doubtful it could withstand peak heat fluxes in the range of 10 kW/cm^2 without spalling. Given these results, it was decided to select fully dense carbon phenolic as the reference TPS in the nose region.

WINDSIDE CENTERLINE (ZONE 2)

The "Low" and "Med" aeroheating rates and loads along Monte Carlo trajectory #1647 are shown in Figure 9. As was the case with the nose region, after further aeroheating analyses, it was concluded that the "High" values were outside of the expected range and the "Med" level of aeroheating was used for TPS sizing for the reference vehicle. The peak heat fluxes are in the range 3-6 kW/cm^2, eliminating lower density candidate materials. The total heat loads are still very large – 300-700 kJ/cm^2. TPS sizing in this region was done for fully dense carbon phenolic, the Genesis concept and reduced density carbon phenolic. As seen in Figure 10, the reduced density carbon phenolic has the lowest areal weight (≈ 6.5 g/cm^2 at the reference condition), but is very thick (≈ 6.2 cm). The surface recession is the largest among the materials considered (≈ 3.9 cm). The Genesis concept's areal weight is marginally greater (6.9 g/cm^2) and the thickness is very large (≈ 8.7 cm). The sizing analysis determined that the C-C facesheet needed to be ≈ 3.1 cm thick backed by ≈ 5.6 cm of carbon fiberform. However, the surface recession is the smallest among the materials considered (≈ 2.0 cm). In comparison, fully dense carbon phenolic is the heaviest solution (8.9 g/cm^2 at the reference condition). Moreover, the TPS requirements at the low condition are more than half those at the reference condition. Given these results, the reduced density carbon phenolic was selected as the reference TPS in Zone 2. The major uncertainty is whether reduced density carbon phenolic can handle these peak heat fluxes reliably.

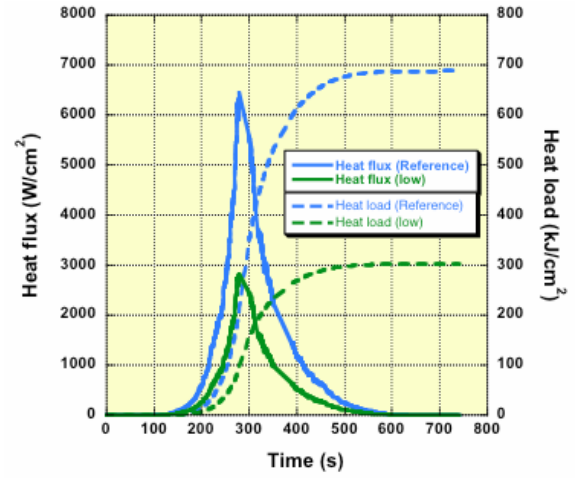

Figure 9. Total low and medium (Reference) heat rates and heat loads for windside centerline region (Zone 2).

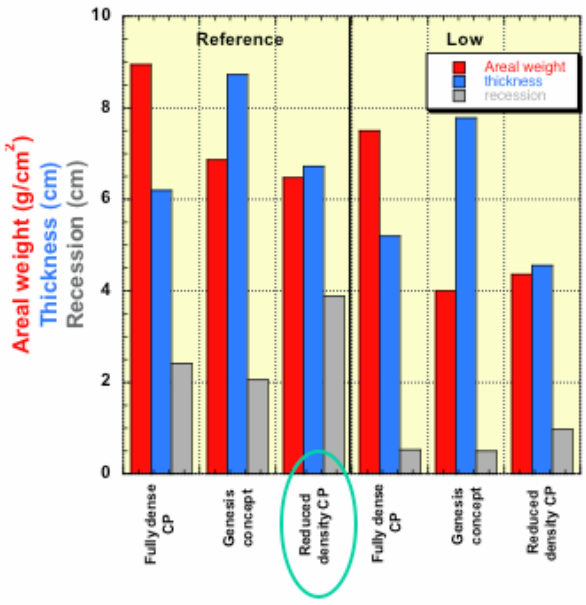

Figure 10. TPS sizing along the windside centerline Zone 2).

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LEE SIDE (ZONES 3 AND 4)

For the lee side of the vehicle, the "Low", "Med", and "High" aeroheating rates and loads along Monte Carlo trajectory #1647 are shown in Figure 11. The "High" values are almost the same as the "Med" values and were eliminated from further consideration. The "Med" level of aeroheating was used for TPS sizing in Zone 3, just aft of the nose on the lee side. Peak heating rates in that region are ≈ 625 W/cm^2 and the total heat load is ≈ 68 kJ/cm^2.

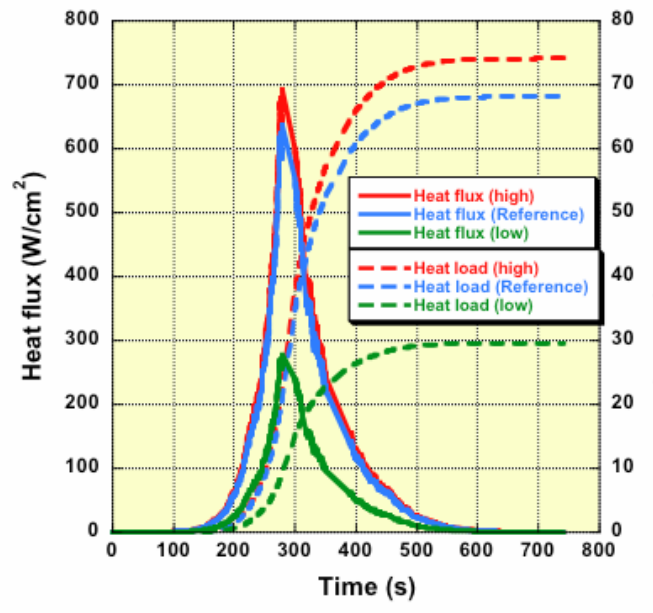

Figure 11. Total heat rates and heat loads on the lee side (Zones 3 and 4).

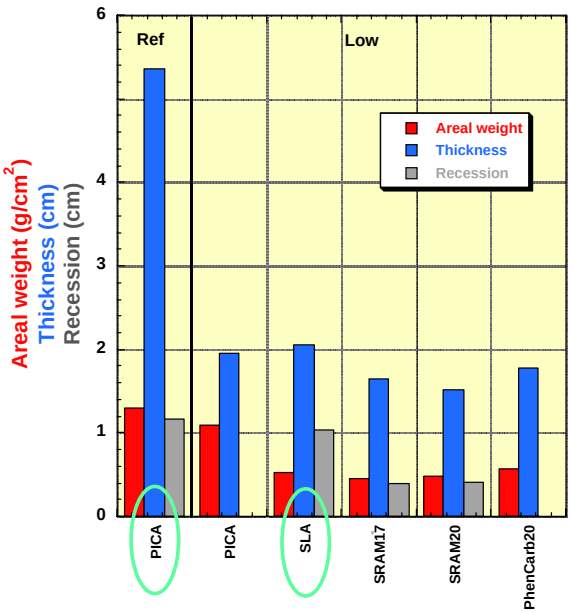

Figure 12. TPS sizing on the lee side (Zones 3 and 4).

Unfortunately, there are not many low-density TPS materials that can handle heat fluxes of that magnitude efficiently. The lack of low-mid density materials with such a capability is a serious deficiency in the TPS arsenal. Consequently, the only candidate TPS material evaluated for Zone 3 applications was PICA (Phenolic Impregnated Carbon Ablator), with a density of ≈ 0.240 g/cm^3.

The PICA sizing results, shown in Figure 12, illustrate that the areal weight is moderate (≈ 1.29 g/cm^2). It is still rather thick (≈ 5.36 cm) but within manufacturing capabilities. Surface recession is predicted to be comparatively small (≈ 1.17 cm).

The low heat flux heating solutions were applied to Zone 4, where the peak heat flux is only 272 W/cm^2 and the total heat load is ≈ 30 kJ/cm^2. In this region TPS sizing calculations were done for PICA, SLA-561V, SRAM17, SRAM20, and PhenCarb20. As seen, PICA would be a heavy and thick TPS solution for this relatively modest heating environment, although no surface recession is predicted. The difference in areal weight and thickness requirements among SLA-561V, SRAM17 and SRAM20 is not negligible, but still relatively small. Predicted surface recession is greatest for SLA-561V. In comparison, no recession is predicted for PhenCarb20 and its areal weight and thickness requirements are similar to the silicone-based low-density ablators. The selection of SLA-561V for the reference design was due solely to its flight heritage.

SUMMARY

This study of a Neptune aerocapture mission has demonstrated that for the geometry and mission design parameters selected, it is a very challenging mission from the TPS standpoint. The windside heating rates are so high (particularly in the stagnation region) that only fully dense carbonaceous TPS materials can handle them with reliable performance. Furthermore, the windside heat loads are enormous. These long duration aerocapture

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trajectories require extremely thick TPS, particularly on the wind side. Such carbonaceous materials are not good insulators, but are good ablators, and the results demonstrate that, on the windside, ablation is the dominant mechanism. Since carbon is the best ablator we currently have, whether improvement could be attained with development of new TPS materials is debatable. However, the required thickness of a tape-wrapped carbon phenolic and/or a cloth-reinforced carbon-carbon cannot be manufactured as a high quality product. It may be possible to manufacture as a stackup of 2.5-5.0 cm thick sheets, mechanically joined. However, that would require some manufacturing development. It is also important to appreciate that the effects of ablation and shape change on vehicle aerodynamics were not considered and should be investigated, as they may be very important.

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